# Section E. Mission Implementation Plan

This section addresses the implementation of the LISA science requirements presented in Section D. The LISA mission builds on a solid foundation of gravitational wave research. It begins with a measurement concept rooted in flight accelerometers, fundamental measurement technologies, and the extensive investment in ground-based gravitational wave systems. The robust LISA architecture is the result of 20 years of studies involving extensive experimental research (see Figure E-1).

The LISA Team has a proactive risk management approach that is implemented within system engineering, addresses the risks associated with all performance requirements, and includes the risks within the technology development effort. The top technical risks are identified, assessed, and mitigated as appropriate.


**Figure E-1: The LISA mission builds on a solid foundation of gravitational wave research**

## E.1 Mission Overview

**The LISA mission design is straightforward. We have ample mass and power margins, low data rates, minimal command and control requirements, and no need for orbit maintenance or station-keeping.**

The LISA Observatory consists of an array of free-drifting proof masses (two per spacecraft) enclosed in three identical spacecraft, where the relative displacement between proof masses is measured using laser interferometry. The spacecraft is a short cylinder with a diameter of 2.7 meters, a height of 0.9 meters, and a mass of approximately 400 kg. The spacecraft bus design is easily implemented with readily available, space-qualified hardware.

The Mission Operations Center (MOC), located at JPL, communicates with the spacecraft via the Deep Space Network (DSN). Science data is forwarded from the MOC to the science data analysis center for calibration and archiving, template-fitting, and source cataloging.

Other key features include:

- Single Delta IV-class launch for all three spacecraft with a Launch Readiness Date (LRD) in 2011
- Thirteen month cruise phase
- Five year science operation requirement, ten year goal
- Heliocentric orbit at 1 AU, following Earth by 20°
- 7 kbps downlink rate / 2 kbps uplink rate
- Single-mode science operations (no targeting, minimum scheduling)
- No ground intervention during nominal science operations

NASA is responsible for the overall flight and ground software architecture. This architecture provides for on-board fault detection, attempted on-board reconfiguration, and spacecraft safing functions for each of the three spacecraft. NASA is also responsible for the integration, test, and verification of the end-to-end flight and ground system.

The challenges to LISA lie in the payload implementation and performance testing. LISA has a comprehensive technology development effort that addresses both of these key issues (see Section F). We have defined an Integration & Test (I&T) flow to support the allocation of payload development responsibilities.

Foldout E-1 provides an overview of the mission.

# E.2 General Information - Mission Design

## E.2.1 Launch and Separation

The mission begins with a single launch of all three spacecraft in a Delta IV-class vehicle. The initial sequence of mission events is common to many planetary missions: ascent, fairing separation, circularization burn, coast, spin-up, Earth-escape trajectory burn, de-spin, and upper stage separation.

Initially, the spacecraft are powered from the launch vehicle umbilical and their own batteries. After stage separation, the three spacecraft power up, separate from each other, perform self-diagnostics, and enter safe-mode with the solar arrays oriented to the Sun and the communications system waiting for ground contact via the low gain antennas. After ground contact has confirmed spacecraft health and trajectory, final course command sequences are uploaded. Each spacecraft and its attached propulsion module then initiates the cruise maneuvers.

## E.2.2 Cruise Trajectory

The transfer from Earth to the operational orbit lasts approximately 13 months. After separation from the upper stage, the three spacecraft are already in Earth-escape trajectories. Each spacecraft is commanded to perform an initial, mid-course, and final insertion burn to its respective operational orbit. DSN contacts for health and safety checks, ranging, and navigation are performed every few days, using the low gain antennae.

The three spacecraft arrive at their respective orbits within a month of each other. Foldout E-1 depicts the Earth-Sun centered trajectory plot for all three spacecraft.

The final orbit insertion maneuvers are performed after DSN Doppler and ranging measurements have determined orbit position to within 10 km and inter-spacecraft range rate to within 10 meters/second (m/s), as required for the operational orbit. This precision is easily met by current DSN capabilities. The propulsion module is jettisoned after final orbit insertion.

## E.2.3 Operational Orbit Information

In their operational orbits, the three spacecraft are at the vertices of an equilateral triangle, nominally 20º following the Earth. Each side of this triangle (“d” in Table E-1) is 5 million km, as shown in Figure E-2.

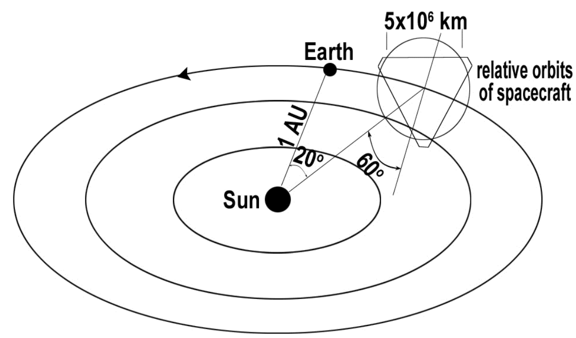


**Figure E-2: LISA Formation. Three spacecraft in an equilateral triangle.**

The above orbital configuration is defined by the orbital elements described in Table E-1.

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***"LISA has a mission architecture that is simple and robust - with ample margins"***


## Spacecraft Characteristics

| | |
|---|---|
| **Implementation** | Drag-free proof masses, distance changes measured with interferometer |
| **Observatory** | 3 identical spacecraft in equilateral triangle, $5 \times 10^6$ km arm length |
| **Orbit** | Heliocentric, 1 AU, 20° behind Earth |
| **Spacecraft Power** | 432W |
| **Spacecraft Dimensions** | 2.7 m diameter by 0.89 m height |
| **Spacecraft Mass** | 400kg |
| **Propulsion Module Mass** | 828kg |
| **Launch Mass** | 3562kg |
| **Launch Vehicle** | Delta IV 4240 or equivalent |
| **Mass Margin** | 30% reserve plus 11% mass margin |
| **Performance Margin** | Factor of 10 on disturbance accelerations and measurement sensitivity |
| **Telemetry** | DSN X-band, 7kbps |
| **Data Volume** | 40 Gbytes |
| **Science Operations** | 5 years nominal, 10 years extended |

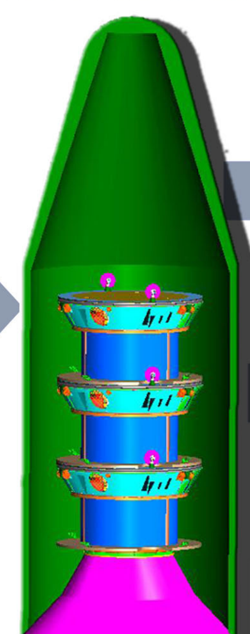
## Timeline

**MISSION LIFETIME GOAL - 10 YEARS**

1st Week — 1st Year — 5 Years

### Launch and Early Orbit

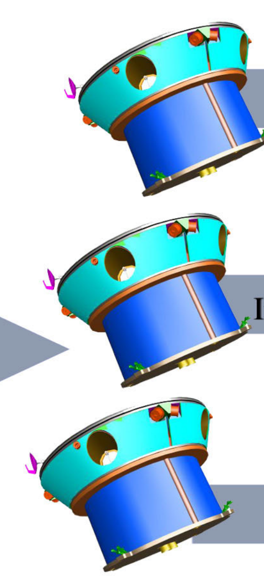
Delta IV Medium +(4, 2)



Separation Checkout

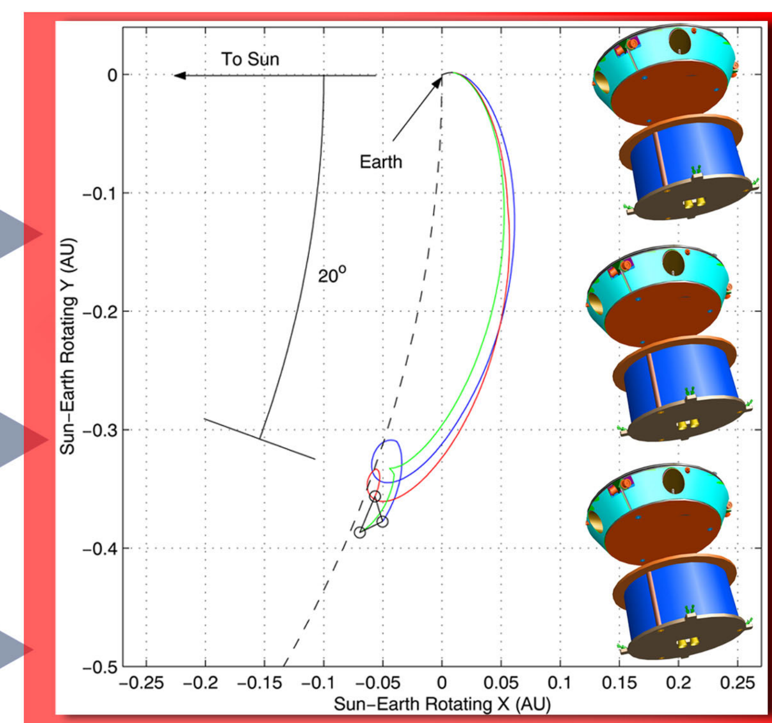
Transfer Orbit Burn

### Cruise Phase



Orbit Insertion

### Deploy and Activate




Separate Propulsion Modules

Instrument Checkout

Acquire Interspacecraft Link

Science Commissioning

### Normal Science Operations

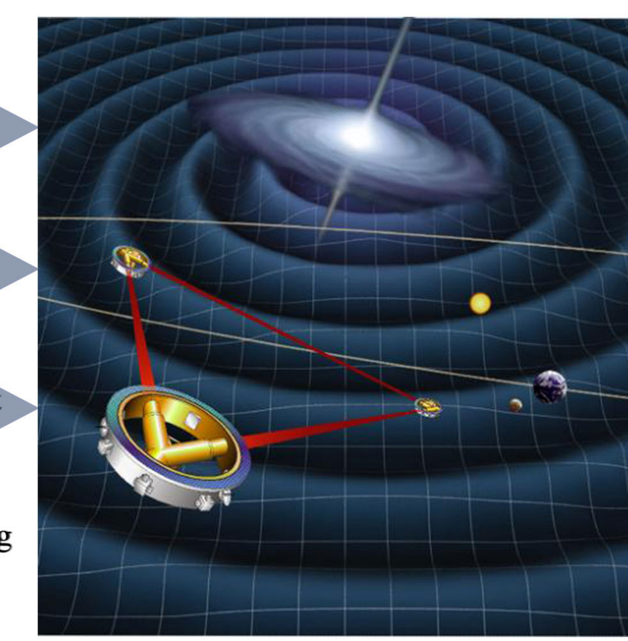


**Table E-1: LISA Operational Orbit Elements, Earth Mean Ecliptic 2000 coordinates**

| | | | |
|---|---|---|---|
| Semi-major axis, a = | 1 AU (149.6 million km) | | |
| Eccentricity, e = | d / (2 a √3), (9.65%) | | |
| Inclination with respect to the ecliptic, I = | d / (2 a), (0.0167) | | |
| Argument of perihelion = | 90° or 270° from Aries | | |
| The ecliptic longitude of the ascending node, Ω, and the mean anomaly, M, of the three spacecraft differ by 120°: | | | |
| | S/C 1 | S/C 2 | S/C 3 |
| Longitude of ascending node | Ω | Ω+120° | Ω-120° |
| Mean anomaly | M | M-120° | M+120° |

These orbits provide stability and adequate solar power over the full ten-year mission life goal, and permit DSN communication with robust link margins. The three spacecraft 'pin-wheel' about a point centered approximately on the ecliptic. There is no orbit maintenance required and the only propulsion requirement is drag-free control.

## E.3 Payload

The LISA science instrument consists of the three spacecraft, including the payload, operating together. The instrumentation concept is described in Section D. Payload implementation is described in this section, and the spacecraft implementation is described in Section E.4.

The LISA payload design is the product of extensive investment in the U.S. and Europe, and has strong heritage in both ground- and space-based measurement systems.

The payload achieves the science measurement by performing two key functions:

- Reducing disturbance accelerations to the proof masses to less than $3 \times 10^{-15}$ m/s$^2\sqrt{\text{Hz}}$
- Measuring the change in distance between proof masses in free-fall to a sensitivity of $4 \times 10^{-11}$ m/$\sqrt{\text{Hz}}$

These functions are provided by the Disturbance Reduction System (DRS) and the Interferometry Measurement System (IMS). Foldout E-2 provides an overview of the LISA payload and its elements. Most of the payload is housed in two hollow cylinders joined in a 60° 'Y' tube. Additional components are mounted on the spacecraft bus to minimize their thermal and electrical disturbances relative to the proof masses. The payload and spacecraft are designed to minimize spurious gravitational, magnetic, or electrical forces on the proof mass. This requires ultra-stable structures, several levels of thermal isolation, tight control of mass properties, restrictions on magnetic fields and power fluctuations, and cleanliness.

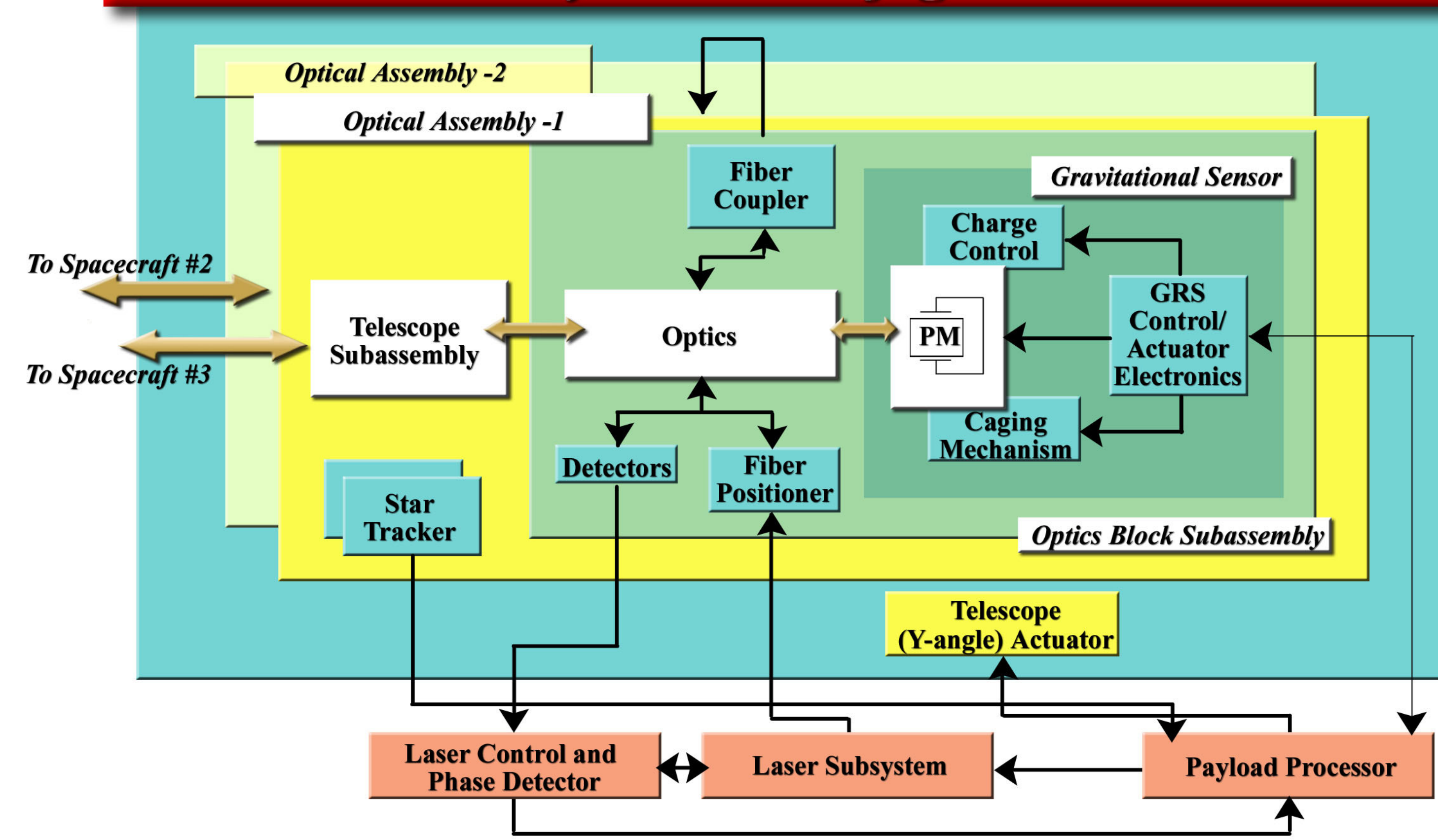
### E.3.1 Disturbance Reduction System

The DRS isolates the proof masses from external disturbances and controls the spacecraft attitude and position to maintain drag-free flight and laser pointing. A non-contact electrostatic suspension system both measures the position of the proof mass along the sensitive axis, and controls the other degrees of freedom (two transverse to the sensing axis and three rotations). The position of the proof mass serves as the input to the drag-free control system, implemented by the payload processor. A simple, low-bandwidth, feedback loop commands the µN thrusters to center the spacecraft about the proof mass on the sensitive axis. Detectors on the optical bench measure the position of the distant spacecraft within the field of view to maintain the laser beam pointing. Table E-2 summarizes the pointing requirements for the payload.

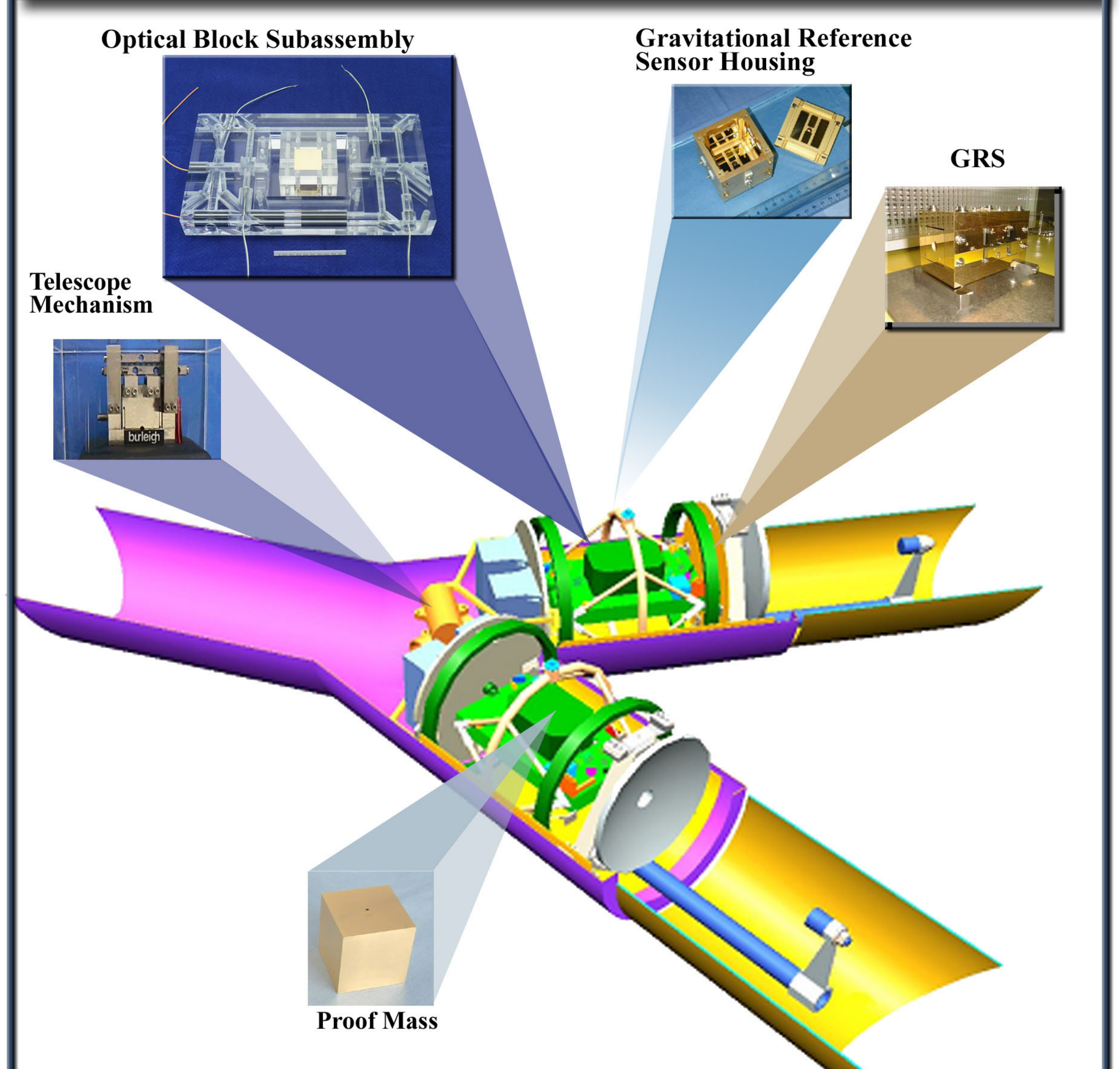
During initial acquisition and any required re-acquisition, the attitude control system can also use information provided by two Ball CT602 star trackers that are bore-sighted with the payload telescopes. A spare star tracker is mounted alongside each as a cold backup. We have identified no


***"LISA's robust payload architecture is the result of 20 years of studies backed by extensive lab analysis"***

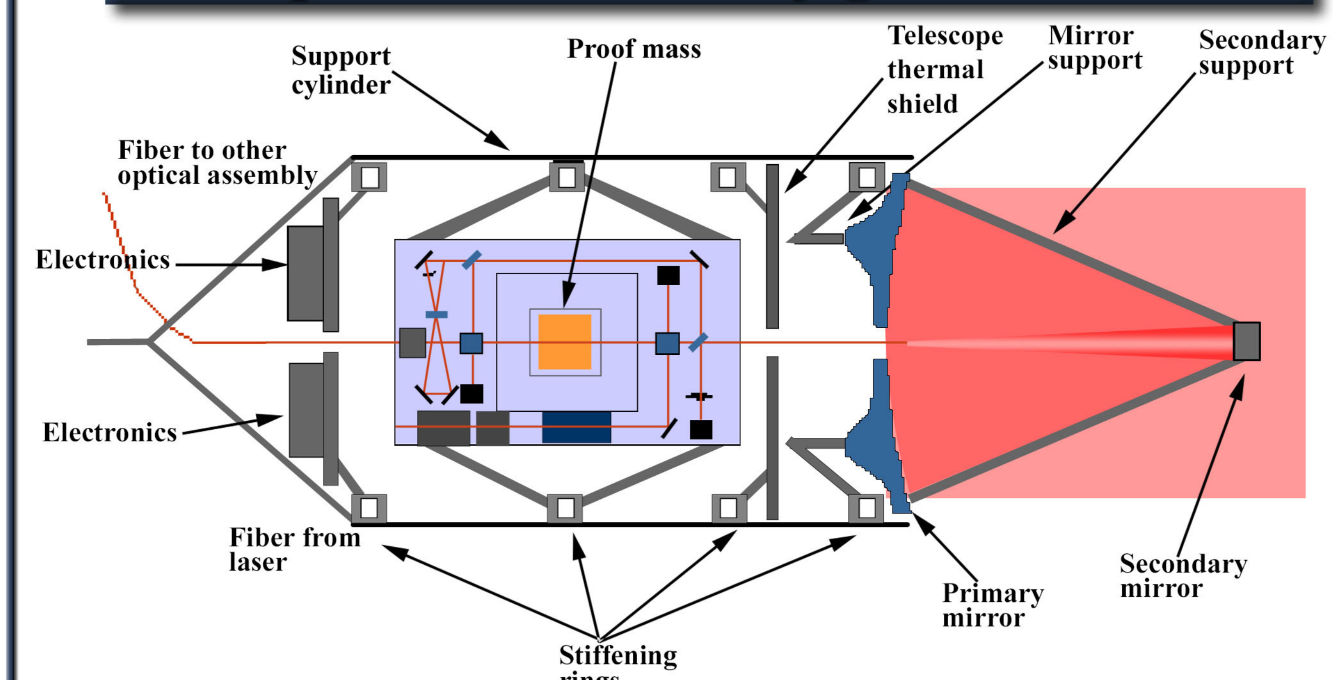
## *Payload Configuration*




## *Payload Y-Tube Assembly*




## *Optical Bench Configuration*




## *Primary Systems*

**Disturbance Reduction System (DRS)**

**Interferometry Measurement System (IMS)**

## *Primary Components*

**DRS**

***consists of gravitational sensor, drag free controller, and micro Newton thrusters - maintains spacecraft position relative to proof masses***

**IMS**

***consists of telescope, optics, laser, and phase measurement subsystem - measure the change in distance between proof masses***

significant risks with the star trackers and currently available commercial units meet the field of view and pointing accuracy requirements.

#### E.3.1.1 Gravitational Reference Sensor

At the heart of the DRS is the Gravitational Reference Sensor (GRS), including the proof mass and housing. The GRS measures the proof mass position while minimizing coupling to spacecraft motion. The controller maintains drag-free motion along the sensitive axis within the Measurement Bandwidth (MBW). Sensing electrodes on the walls of the vacuum enclosure measure the proof mass position by capacitance changes. The same electrodes are also used to apply electrostatic control forces to the proof mass as required along the non-sensitive axes. For the pre-launch and cruise phases of the mission, a mechanism cages the proof mass to protect it. A charge control system uses ultraviolet light to neutralize any charge buildup on the proof mass.

The major risk of the GRS is that it may not meet the acceleration noise requirement. This prompted three parallel technology development activities, discussed in Section F.2. Two of these paths produce a flight-qualified GRS before Phase C/D. The third path studies disturbance effects that may limit performance such as patch field effects and thruster noise. Laboratory measurements already show acceleration noise performance within one order of magnitude of the minimum mission.

#### E.3.1.2 Micro-Newton (μN) Thrusters

The key actuator of the DRS is the low-force thruster used to accomplish fine control of the spacecraft position and attitude to maintain drag-free flight. LISA uses Field Emission Electric Propulsion (FEEP) thrusters, first developed in the mid-1970s. FEEPs have a high specific impulse (6,000 to 10,000 seconds) and use high voltage to accelerate metallic ions. Each FEEP consists of a metal reservoir, a needle array or slit emitter, an electron emitter to neutralize the plasma, and supporting high voltage electronics. They are arranged in clusters of four at three different points on the spacecraft, and provide redundant control in all six degrees of freedom.

The key risk is that the thrusters may not exhibit adequate lifetime. A minor risk is that they may exhibit excessive noise. However, preliminary laboratory tests show that the thrusters meet our performance noise requirements. Laboratory tests of reliability are underway. Increasing the gain in the DRS control system can compensate for decrements in thruster performance. Lifetime concerns could be addressed by adding additional thrusters and re-allocating mass reserves at the system level.

### E.3.2 Interferometry Measurement System

The IMS, the other major payload element, measures the displacement between a local and a distant proof mass. It consists of the telescope, interferometer optics, the phase measurement system, laser system, and the laser and clock frequency noise correction system.

#### E.3.2.1 Telescope

The LISA telescope is a conventional Dall/Kirkham F/1.4 design. The telescope gathers laser light from the remote spacecraft and focuses it on the interferometer optics. The telescope is made from Silicon Carbide or Ultra-Low Expansion (ULE™), which satisfies the dimensional stability requirement. No new technology is required for telescope development. Performance requirements such as aperture and optical quality can be traded against laser power and pointing requirements.

#### E.3.2.2 Interferometer Optics

Behind the telescope is a thermal shield, then the optical metering structure holding the interferometer optics, laser stabilization cavity, and detectors. The main requirement on all of the optics is high dimensional stability. This is satisfied by a thermally stable design using either ULE™ or Zerodur™. The remaining risk in the optical assembly is that it may exhibit excessive structural noise at low frequencies ($\approx 10^{-4}$ Hz). Testing is underway to gather design data in Formulation (see Section F.2.1).

### E.3.2.3 Phase Measurement System

The phase measurement system measures phase changes on the order of $10^{-6}$ cycles/ $\sqrt{\text{Hz}}$ in the presence of a Doppler shift of several MHz. The design, which has evolved along with TDI and transponder ranging techniques, draws from established technologies, including those employed in the Global Positioning System (GPS). The primary components are an ultra-stable oscillator (USO) as a frequency reference and the phase meter. The USO stability requirements have been demonstrated on previous missions, including Mars Observer. The beat signal from the photodiode on the interferometer optical bench is beat against a comb of frequencies in order to remove the Doppler shift and produce a final beat frequency on the order of 10 to 100 kHz. Analog electronics shape this signal to produce a train of zero-crossing pulses. The pulse train is fed to a phase meter employing a digital phase-locked loop, similar to that used in GPS receivers.

In an alternative approach, the number of pulses between USO clock pulses is counted to measure the arm path length variation. This digital count train is stored and processed further by the noise reduction system. The technology development described in Section F.2.2.2 is motivated by the need to mitigate the risk that the phase measurement system will not meet the performance requirements. Laboratory instruments have already been developed which approach these requirements. Further work focuses on improving bandwidth and sensitivity.

### E.3.2.4 Laser System

LISA requires 1 W lasers with high frequency stability ($< 30$ Hz/$\sqrt{\text{Hz}}$) that can be tuned over 10 GHz. Diode-pumped Non-Planar Ring Oscillators (NPRO), as shown in Figure E-3, have already demonstrated this performance in the laboratory, at nearly the required power levels.

Detectors on the optical bench provide the feedback for power control, while a temperature stabilized cavity controls the frequency. A polarization preserving optical fiber delivers light from the laser assembly mounted on the lower spacecraft plate. The fiber positioner that conducts the laser light onto the optical bench also provides the switching function between the primary and backup laser.

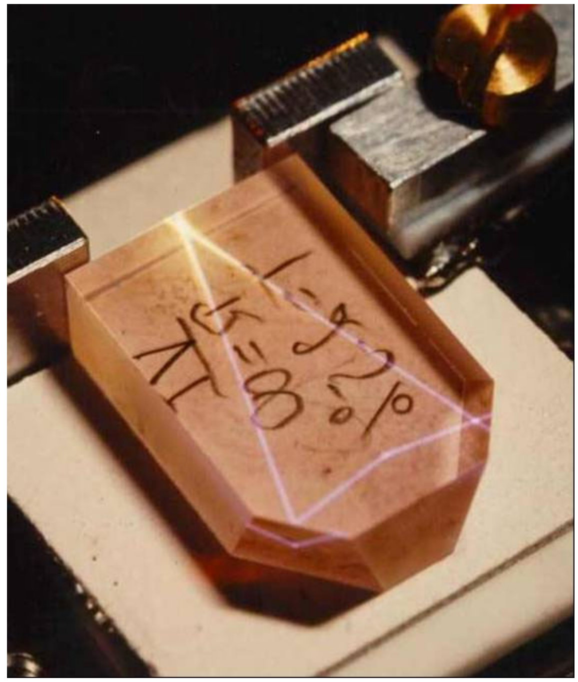


**Figure E-3. NPRO Nd-Yag Crystal**

The primary risk in the laser system is inability to space-qualify the lasers to operate at the required power for the life of the mission. Three parallel risk mitigation efforts are underway (see Section F.2.2.2). The system design has trade space to accommodate this concern. Each payload assembly has a spare laser and the design permits de-rating the power level to prolong on-orbit life. Increasing the telescope aperture or shortening the arm distance can be traded to restore signal strength. Should it prove necessary, the Project holds mass margin for additional spare lasers.

### E.3.2.5 Laser and Clock Frequency Noise Suppression

The remaining noise in the laser and clock is suppressed several orders of magnitude by the TDI technique implemented on the payload processor. This method requires knowledge of the three inter-spacecraft distances that vary in a predictable way once the orbits are established. Direct measurements are made periodically using the ranging tones imparted by the electro-optic modulator.

The risk of the TDI technique is that the noise may be inadequately suppressed. Experimental and analytical studies are underway to address this concern. Range measurement accuracy, currently 200 m, can be traded off against other noise sources addressed by TDI, including laser frequency and clock stability.

### E.3.3 Payload Structural Subassembly

Besides the payload items physically mounted on the lower spacecraft plate, the payload is completed by two other components, the Y-tube and the Telescope Actuator.

#### E.3.3.1 Y-Tube

The optical assemblies are mounted in two hollow cylinders, joined in a 60° ‘Y’-shaped structure that provides mechanical support and thermal isolation from the spacecraft. This structure is composed of graphite-epoxy, gold coated on both the interior and exterior to reduce radiative heat transfer. The ultra-stable oscillators, part of the phase measurement system, are also mounted on the back of the Y-tube, behind a thermal shield. There are no significant risks associated with this structure.

#### E.3.3.2 Telescope Actuator

The inter-spacecraft motion causes a ±1° variation in look angle to the remote spacecraft. An actuator on each optical assembly makes this pointing correction by changing the angle between the telescopes and optical assemblies from the nominal 60°. The actuator operates in only one axis; the other axis is controlled by rotation of the spacecraft. The mechanism swivels the assembly about the center of the proof mass enclosure. Each mechanism has adequate range to perform the angular correction thus providing redundancy. While similar devices were developed for other projects, LISA requires a lower disturbance level and a greater dynamic range. Detailed analysis is underway to address how these more stringent requirements can be met.

## E.4 Spacecraft

The European-provided spacecraft utilizes a complement of flight-proven subsystems that are well within the state of the art. The spacecraft supports the payload and provides thermal protection, power, communications, command and data handling, fault detection and safing, and attitude and orbit control. A propulsion module provides the propellant and thrusters needed to reach the operational orbit.

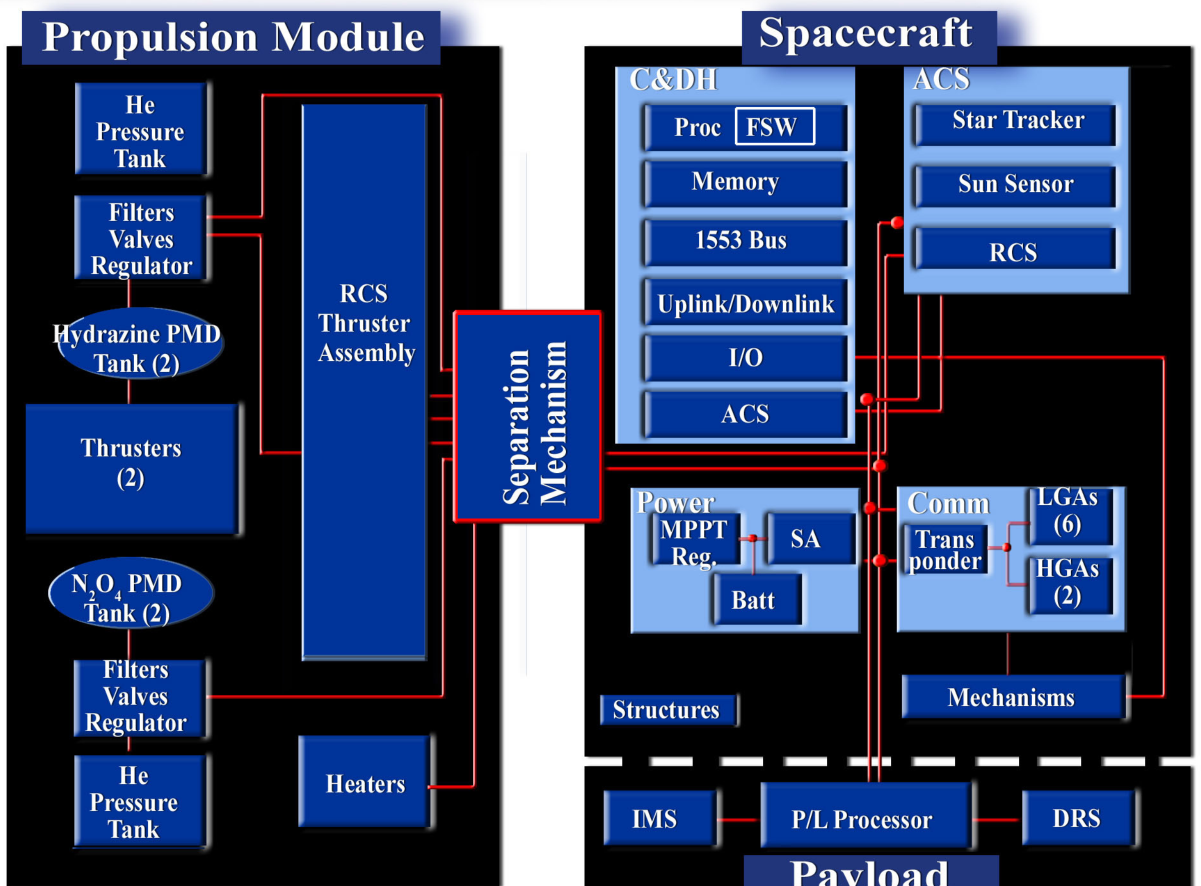
Foldout E-3 shows the LISA spacecraft in its orbital configuration and provides a functional block diagram of the spacecraft. A fixed Gallium Arsenide (GaAs) solar array and Li ion batteries provide power. X-band transponders, low gain and high gain antennas provide communications with the DSN. A standard Command and Data Handling (C&DH) system using a MIL-STD-1553 data bus provides more than enough capacity for the relatively low science data rate, subsystem management, and the handling of housekeeping data. Finally, the attitude and orbital control system commands thrusters to orient the spacecraft during cruise and provide a drag-free environment under science operation.

When the spacecraft are operational, the Sun is always 30° off normal of the sunward face of the spacecraft. This provides a thermally stable environment throughout the spacecraft and especially within the payload. The spacecraft electronics boxes are located well away from the proof mass housings to minimize their thermal and gravitational effects.

### E.4.1 Structures

The spacecraft bus is a graphite epoxy structure that supports the payload during launch and otherwise provides thermal insulation. It consists of two circular decks joined by stiff tubes to form a short cylinder. Hard point restraint mechanisms are mounted on the top and bottom of each tube. The outer ring of the cylinder has openings to accommodate the apertures of the Y-tubes. The top deck of the spacecraft bus has mountings for the solar arrays and high gain antennas. The internal decks house electronics boxes, the payload processor, and the laser system. Three clusters of FEEP thrusters are mounted at the vertices of the internal support structure and extend outside the outer ring of the cylinder.

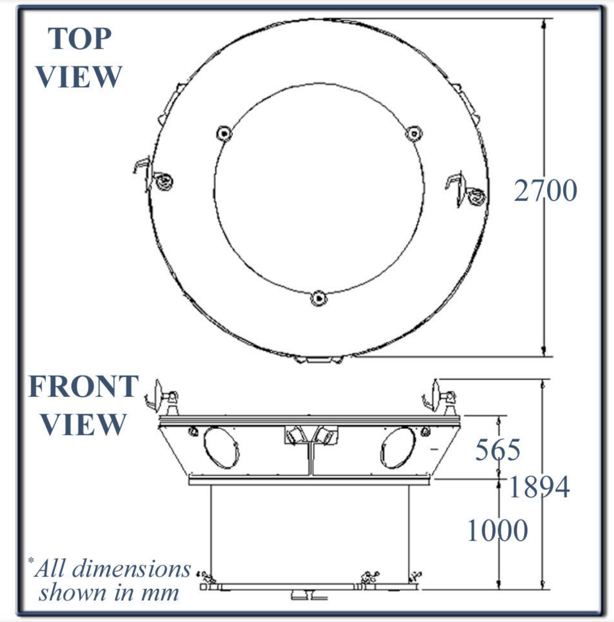
# LISA Flight Segment

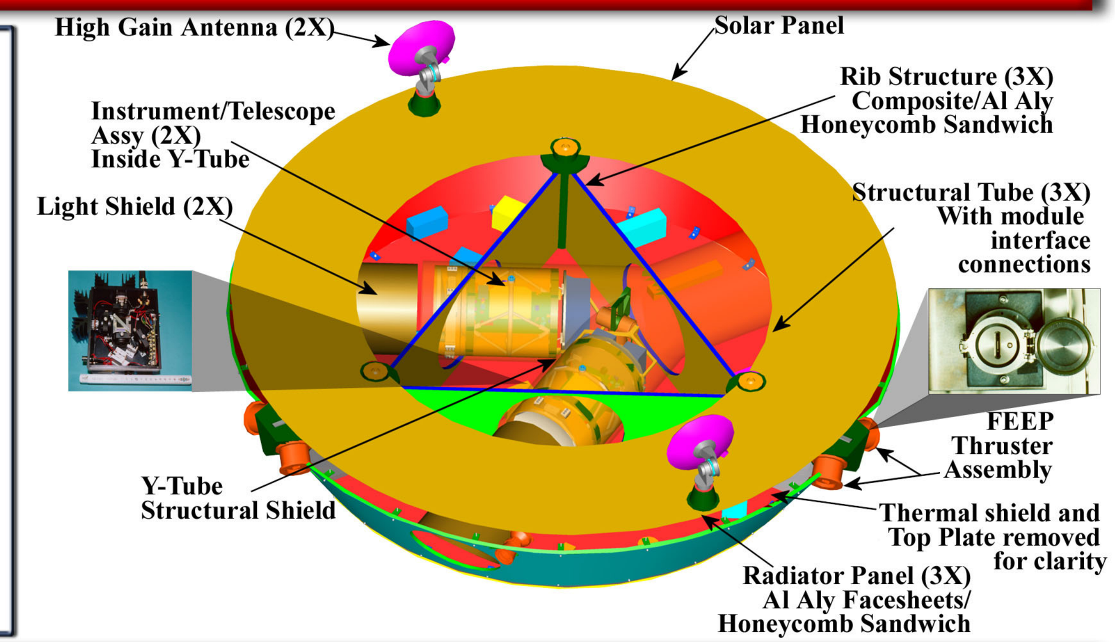



*see foldout E-2 for more detail*

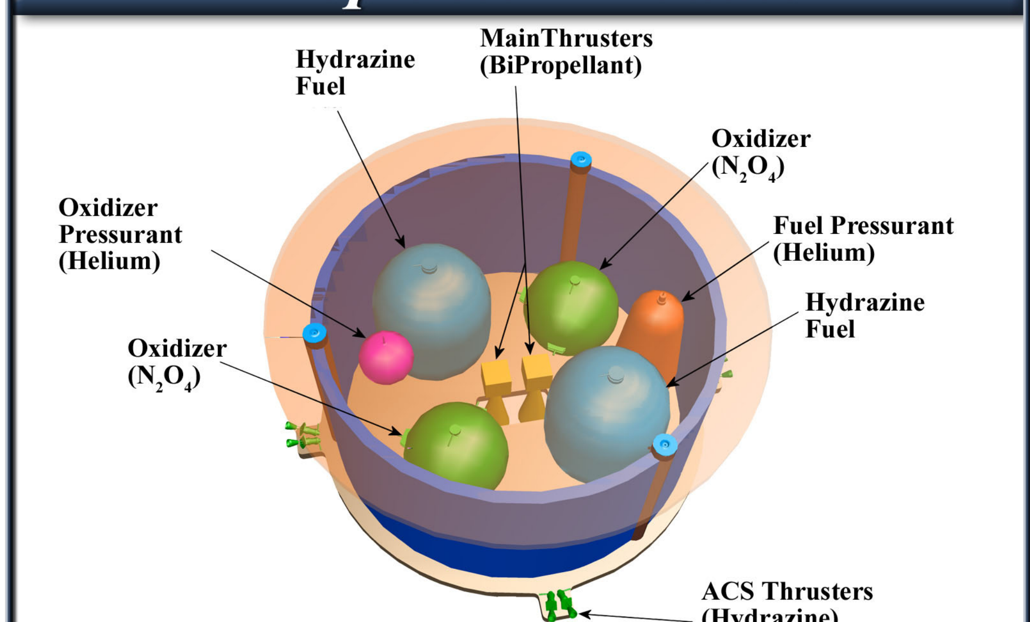
## *"LISA uses proven technology and heritage hardware – with low data volumes and no critical C&DH timing"*

## *Operational Configuration*




## *Propulsion Module*




## *Resources and Margins (Mass and Power Budget)*

| Msn. Element | Sub-sys. | Component | Mass: Each | CBE Total | Mass Growth Cont. (%) | Mass Growth Cont. (kg) | CBE + Contin. | Power: Each | CBE Total | Power Growth Cont. (%) | Power Growth Cont. (W) | CBE + Contin. |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| LISA Flight System (3 Wet Spacecraft and Launch Vehicle Adapter) | | | | 3093.00 | 12% | 384.45 | 3562.31 | | | | | |
| Spacecraft (Spacecraft bus + Payload) | | | Total | 308.97 | 30% | 92.69 | 401.66 | Total | 332.20 | 30% | 99.66 | 431.86 |
| | Structures & Mechanisms | | Total | 75.90 | 30% | 22.77 | 98.67 | Total | 1.00 | 30% | 0.30 | 1.30 |
| | Power | | Total | 28.85 | 30% | 8.65 | 37.50 | Total | 11.00 | 30% | 3.30 | 14.30 |
| | Command & Data Handling | | Total | 15.90 | 30% | 4.77 | 20.67 | Total | 25.00 | 30% | 7.50 | 32.50 |
| | Telecom | | Total | 19.60 | 30% | 5.88 | 25.48 | Total | 85.00 | 30% | 25.50 | 110.50 |
| | Attitude Control | | Total | 18.52 | 30% | 5.56 | 24.08 | Total | 48.60 | 30% | 14.58 | 63.18 |
| | Thermal Control | | Total | 11.00 | 30% | 3.30 | 14.30 | Total | 0.00 | | 0.00 | 0.00 |
| | Cabling | | Total | 21.00 | 30% | 6.30 | 27.30 | Total | 0.00 | | 0.00 | 0.00 |
| Payload | | | Total | 118.20 | 30% | 35.46 | 153.66 | Total | 161.60 | 30% | 48.48 | 210.08 |
| | Optical Assembly | | Total | 66.30 | 30% | 19.89 | 86.19 | Total | 124.60 | 30% | 37.38 | 161.98 |
| | Gravitational Reference Sensor Assy | | Total | 19.00 | 30% | 5.70 | 24.70 | Total | 11.00 | 30% | 3.30 | 14.30 |
| | Y-Tube Assembly | | Total | 15.00 | 30% | 4.50 | 19.50 | Total | 1.00 | 30% | 0.30 | 1.30 |
| | Thermal Control | | Total | 2.00 | 30% | 0.60 | 2.60 | Total | 0.00 | | 0.00 | 0.00 |
| | Payload Processor | | Total | 15.90 | 30% | 4.77 | 20.67 | Total | 25.00 | 30% | 7.50 | 32.50 |
| Propulsion Module | | | Total | 212.49 | 30% | 63.75 | 276.24 | Total | 264.00 | 30% | 79.20 | 343.20 |
| | Structure | | Total | 101.00 | 30% | 30.30 | 131.30 | Total | 0.00 | | 0.00 | 0.00 |
| | Chemical Propulsion | | Total | 85.40 | 30% | 25.62 | 111.02 | Total | 98.00 | 30% | 29.40 | 127.40 |
| | ACS | | Total | 22.09 | 30% | 6.63 | 28.72 | Total | 131.00 | 30% | 39.30 | 170.30 |
| | Thermal Control | | Total | 4.00 | 30% | 1.20 | 5.20 | Total | 35.00 | 30% | 10.50 | 45.50 |
| | Propellant | | Total | 1359.00 | 0% | 0.00 | 1359.00 | Total | 0.00 | 0% | 0.00 | 0.00 |
| | L/V Adapter (5%) (kg) | | Total | 169.63 | 0% | 0.00 | 169.63 | Total | 0.00 | 0% | 0.00 | 0.00 |
| | Spacecraft Mass (kg) | | | 926.90 | 30% | 278.07 | 1204.97 | Payload Power | 161.60 | 30% | 48.48 | 210.08 |
| | Prop Module Mass (kg) | | | 637.47 | 30% | 191.24 | 828.71 | S\C Bus Power | 170.60 | 30% | 51.18 | 221.78 |
| | Dry Flight System Mass (kg) | | | 1564.37 | 30% | 469.31 | 2033.68 | Prop + S/C Bus Po | 434.60 | 30% | 130.38 | 564.98 |
| | Propellant (kg) | | | 1359.00 | 0% | 0.00 | 1359.00 | Spacecraft Power | 332.20 | 30% | 99.66 | 431.86 |
| | Wet Flight System Mass (kg) | | | 2923.37 | 16% | 469.31 | 3392.68 | | | | | |
| | TOTAL LAUNCH MASS (kg) | | | 3093.00 | 12% | 384.45 | 3562.31 | Spacecraft Power with 8% loss in wiring harness | | | | 466.41 |
| LAUNCH VEHICLE CAPABILITY | | 4003 | | | | | | | | | | |
| Delta 4 Medium - +(4, 2) | Performance Margin (kg) | 441.00 | | | | | | Square Meters of Array needed (EOL) | | | | 3.45 |
| | % | **11.00%** | | | | | | (for cruise phase (prop + s/c bus) with 8% line loss) | | | | |
| LAUNCH VEHICLE CAPABILITY | | 4516 | | | | | | Square Meters of Array needed (EOL) | | | | 2.64 |
| Delta 4 Medium - +(5, 4) | Performance Margin (kg) | 954.00 | | | | | | (for operations phase (Spacecraft bus + payload) power with 8% line loss) | | | | |
| | % | **21.00%** | | | | | | | | | | |

### E.4.2 Telecommunication

The telecommunication subsystem meets the modest 7 kbps downlink / 2 kbps uplink requirements with a Radio Frequency (RF) package comprised of two high gain antennas, six low gain antennas, an X-band transponder, and associated electronics. The high gain antennas provide complete 360º coverage. In order to minimize interruptions of the science data, all six antennas are used for four days each before all of them are rotated together. Data is normally downlinked every other day, with all the data being sent by the laser links to a designated master spacecraft. The six low gain antennas are used during cruise and/or emergency modes (see Table E-4).

### E.4.3 Thermal

The LISA spacecraft employs a completely passive thermal design. No active control or heaters are required. Various layers of thermal isolation provide the required thermal performance. The first layer is the insulation between the solar array panel and the top plate. A second layer is the decoupling between the spacecraft structure and the Y-tube. Inside the Y-tube, a gold-plated thermal shield surrounds the optical bench. At the telescope end of the tube is a thermal shield between the optical bench and the primary mirror, and at the opposite end is a plate to provide isolation between the digital / analog electronics located at the intersection of the Y-tube arms.

### E.4.4 Power

The power system requirements are met by a conventional design with a GaAs solar array and a 9 Ah Li-Ion battery. The bus regulator is a Maximum Power Point Tracking (MPPT) design that supports an orbital average load of 432 W at end-of-life (including 30% contingency). The 3.45 $m^2$ solar array is body-mounted on the top plate of the spacecraft and sized to support the cruise stage peak power requirement of 565 W. The bus is regulated at 28 V ± 1%. The detailed power budget including contingency is on Foldout E-3.

### E.4.5 Attitude Control

The Attitude Control System (ACS) maintains pointing during the cruise and safe modes and monitors attitude during science-mode. The spacecraft ACS has modest requirements; the more stringent pointing requirements are the responsibility of the payload. Table E-2 summarizes the ACS control requirements.

**Table E-2: ACS Control Requirements**

| Pointing Requirement | Value (all values are 3 sigma per axis) |
|---|---|
| Pointing accuracy | Spacecraft (1º); Payload (0.9 micro-radian relative to received laser) |
| Pointing knowledge | Spacecraft (1º); Payload (0.9 micro-radian relative to received laser) |
| Pointing stability | Spacecraft (1º); Payload (8 nano-radian/√Hz in LISA measurement band width) |
| Bias | Star tracker to telescope alignment (post-calibration) < 225 micro-radian |
| Drift | No requirement |
| Scan/Agility | There are no scan-like or agile operations |
| On-orbit calibration | Star tracker to telescope alignment calibration is made once (during commissioning) |
| Attitude knowledge processing | Real-time, on orbit, from star tracker data. Alignment drift post-processed on ground |
| Deployments | Propulsion module jettison, HGA setup, proof-mass uncaging. |
| Articulations | The two HGAs are steerable (±180º). The telescopes articulate in one axis (1º). |

In cruise, the spacecraft ACS uses sun sensor and star tracker data to command the thrusters of the propulsion module to maintain a coarse pointing (1°) in three axes. The spacecraft ACS continuously monitors the attitude and enters safe mode when the sun angle error exceeds a threshold value. Safe-mode maintains solar illumination and High Gain Antenna (HGA) pointing to Earth using the µN thrusters. The spacecraft ACS sensors and actuators are redundant. The sun sensors and star trackers produce the required attitude data even if one of the sensors has failed. The µN and propulsion module thrusters are also robust to multiple failures.

### E.4.6 Command and Data Handling

The C&DH function is provided by commercially available spacecraft avionics. The interface to the electronics is implemented using a MIL-STD-1553 data bus. The processing capacity required is on the order of a few thousands of floating point operations per second. Many previously flown spacecraft processors can easily satisfy these requirements.

### E.4.7 Flight Software

There are two processors on each spacecraft; the spacecraft bus processor and the payload processor. ESA is providing the spacecraft processor and software; NASA is providing the payload processor and software. The flight software (FSW) for each spacecraft is identical except for unique factors, such as the ID. NASA is responsible for the overall software architecture and the definition, integration, test, and verification of the interfaces between the processors.

The spacecraft FSW controls all fault detection and safing, C&DH, ACS, electrical, thermal, communications, and related functions on each LISA spacecraft. The payload FSW supports all payload functions including laser acquisition, payload safing, and fine attitude control during science operations.

The integrated software architecture uses Interface Control Documents (ICDs) to define the interfaces between the spacecraft and payload processor, inter-spacecraft communication, and between the three spacecraft and the ground.

### E.4.8 Propulsion Module

LISA employs traditional propulsion modules to transport each spacecraft to their operational orbits. Each propulsion module incorporates a cruise propulsion system and a Reaction Control System (RCS). The cruise propulsion system is responsible for the transfer phase to the operational orbits. The RCS provides three-axis control. The total delta-V for both systems is 1.22 km/s$^2$. The cruise propulsion system consists of two redundant bipropellant ($N_2 0_4$/hydrazine) engines, each providing 22N of thrust through the center of gravity. The RCS employs eight small 0.1N hydrazine thrusters, and shares a common hydrazine supply with the cruise propulsion system.

## E.5 Resources and Margins

The system design for LISA has adequate margins in the key performance areas, including: phase measurement noise, disturbance acceleration noise, spacecraft mass, and electrical power. The System Engineering Team regularly tracks these parameters for early identification of technical problems.

### E.5.1 Spacecraft Mass and Power Margins

The Delta IV launch vehicle has adequate mass margin for launching all three spacecraft including propulsion modules. The detailed mass requirements are given in Foldout E-3. The total launch mass, including the three spacecraft, propulsion modules, propellant and launch vehicle adapter, is 3562 kg including a 30% mass reserve on each subsystem. The selected Delta IV Medium + (4,2) provides an additional 11% margin over the above 3562 kg.

The solar array is sized for operation during the cruise mode. The 3.45 m$^2$ solar array provides 565 W. The power requirement for the spacecraft including the payload, during the science mode is 432 W, including a 30% reserve. The detailed power budget is shown on Foldout E-3. The current design shows the solar array mounted on approximately 60% of the top plate of the spacecraft. If the need arises, approximately 90% of the top plate of the spacecraft can be populated with solar arrays thus providing a 50% margin.

### E.5.2 Performance Margin

Science performance is limited by the spurious acceleration noise acting on the proof mass, and the residual error in phase measurement. The allocated values for these (reference Foldout D-1) are considered within current or near-term state of the art. Further, these allocations are a factor of 10 above the values required for the minimum science mission (reference Figure D-7), leaving a large performance margin at the overall system level.

A contributor to the phase measurement error is pointing stability. Simulations show performance a factor of two better than the required 24 nanoradians (nrad)/√Hz.

## E.6 Assembly, Integration & Test

The integration and test effort ensures that the payload, spacecraft, and observatory are built to satisfy the LISA technical requirements that flow down from the science requirements. The I&T flow is divided into three separate activities involving the integration of the payload, the spacecraft, and the observatory which includes an end-to-end test of the constellation. GRS performance cannot be fully tested on the ground and must be verified by analysis. We verify the GRS performance through integrated modeling anchored in SMART-2 flight data and the appropriate ground-based measurements. The System Engineering Team sets the environmental margins and test durations in accordance with the GSFC General Environmental Verification Specification (GEVS) and the JPL Environmental Assurance Requirements document. A summary of the I&T flow is shown in Figure E-4.

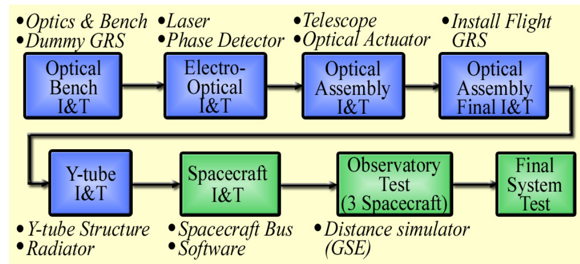



**Figure E-4: The Optical Assembly is the backbone of the I&T flow**

### E.6.1 Payload I&T

The payload integration brings together the first four components listed in Table E-3.

**Table E-3: Progressive I&T Flow Assures Performance Verification and Mitigates Schedule Risk**

| Assembly | Integration Components | Performance Tests | Env. Test |
|---|---|---|---|
| Optical bench | Dummy proof mass, ULE block, fiber, laser stabilization cavity, optics | Wavefront quality, contrast, scatter, bond stability | Vibration |
| Electro-optical Bench | Optical bench, laser, CCD, phase modulator, phase detector, ultra-stable oscillator, electronics | Detector noise, phase noise, laser frequency and amplitude noise, oscillator noise, pointing stability | TV, EMI/EMC |
| Initial Optical Assembly | Electro-optical bench, telescope, star trackers | Wavefront quality, pointing actuation and stability, phase stability | Vibration, Thermal, EMI/EMC, Magnetic |
| Final Optical Assembly | Initial optical assembly, charge management unit, flight proof mass, proof mass housing, caging and vacuum assembly | Charge control, proof mass actuation and control, vacuum level | Vibration, TV, EMI/EMC, Magnetic |
| Spacecraft | Spacecraft bus and payload | Displacement sensitivity, alignment sensitivity, pointing control, wavefront quality, frequency noise, gravity gradient | TV, EMI/EMC |
| Observatory | Three spacecraft | Michelson sensitivity, frequency noise rejection, lock acquisition, spacecraft communication, data processing, constellation testing | Thermal, Vibration, EMI/EMC, TV |

The performance testing in this phase is at the component level, providing confidence in the performance of the entire payload assembly. Environmental testing is also performed throughout this phase.

The full payload testing takes place at JPL after the final optical assembly is integrated with the Y-tube. The interferometric performance of the payloads is tested to the LISA requirements. This is done with the use of a separate test platform that removes seismic disturbances and receives and transmits light to the payload under test. The test platform also includes an optical attenuator and appropriate software to simulate the dispersion and time-delay of the light traversing the $5 \times 10^6$ km arm-length. This tests the full interferometric measurement capability of the payload prior to delivery to GSFC.

### E.6.2 Spacecraft I&T

The spacecraft bus is integrated and tested at ESA, then delivered to GSFC for integration with the payload where the interferometry system performance is again tested. This is done with an external spacecraft simulator similar to that used for the payload testing. The acceleration noise on the proof mass is verified by a combination of test and analysis. In addition, the interface between the payload and spacecraft flight software and the spacecraft to ground system are extensively tested.

### E.6.3 Observatory I&T

The three spacecraft are arranged to test the functioning of the full constellation, with each spacecraft receiving and transmitting light to the other two, incorporating a “range simulator” that provides optical attenuation and emulation of the Doppler shift associated with changes in the on-orbit geometry. This verifies full observatory sensitivity, frequency noise rejection, lock acquisition, observatory communication and data processing, and exercises fault detection and correction capabilities. Operations and end-to-end communication tests are also performed during this phase (see Table E-3).

### E.6.4 Facilities and Equipment

LISA is integrated and verified at JPL and GSFC utilizing existing facilities. All environmental testing is straightforward and there are no system-level calibration requirements. Early integration of the payload by ESA is still in the planning phase.

For payload integration, LISA uses the recently dedicated interferometry laboratory at JPL. This facility – designed expressly for high-precision interferometry systems – provides for subsystem and system level testing of the LISA IMS.

## E.7 Mission Operations and Ground System

The Ground System and operations for the LISA mission are uncomplicated. Key features include autonomous science operations; low data volumes; and use of existing DSN capability and JPL Mission Operations infrastructure. No new technology is needed for the ground system.

### E.7.1 Mission Phases

LISA operations require limited ground commanding and are characterized by long uninterrupted measurement intervals and infrequent configuration changes. The constellation transmits data to the ground autonomously every other day.

During launch, cruise, and orbit insertion, operations focus on navigation and trajectory determination. Once on-station, operations shift to activating the payload, testing all systems for health and performance, and establishing the initial constellation. Then, the on-board controller autonomously sustains science operations, recovers from any anomalies, and enters safe mode in the event of an unrecoverable fault.

During the nominal science phase, there are no interruptions in the inter-spacecraft laser links and no changes to the spacecraft bus or payload configuration except for occasional antenna repositioning. Drag-free operation is autonomously established and maintained to optimize the system availability for science operations. Once science operations have been established, the ground operations are routine, requiring only scheduling of the DSN. There is no need for an around-the-clock Operations Team. The automated ground system alerts an Operations Team member in the event of an anomaly; otherwise, the Operations Team periodically reviews the engineering data to verify nominal spacecraft performance.

### E.7.2 Mission Modes

The following is a list of LISA mission modes:

- ***Launch:*** Begins at launch and ends with the deployment of the three spacecraft
- ***Activation:*** Used for initial on-orbit checkout
- ***Cruise*:** Delivers the three spacecraft to their final orbit insertions
- ***Deployment*:** Separation of propulsion modules from spacecraft
- ***Payload Activation:*** Mechanical release, test, and calibration of payload components
- ***Acquisition:*** Establishes and calibrates the optical links between spacecraft
- ***Normal Science:*** Nominal operating mode
- ***Verification:*** Periodic testing and verification of system capabilities during the normal science mode
- ***Reacquisition:*** Reestablishment of the inter-spacecraft laser links
- ***Engineering:*** Special-purpose diagnostic mode for anomaly investigation
- ***Safe:*** Upon detecting a problem from which the spacecraft cannot autonomously recover

The Acquisition Mode is unique to the LISA mission. The initial inter-spacecraft link acquisition occurs once the ground has checked out the individual payload components. One spacecraft is designated the master. Using position information uplinked from the ground, the master turns on its lasers and scans the predicted location of the slave spacecraft. At the end of the scan (which requires less than two hours) the master stops transmitting and the slave turns on its laser in the direction where it detected the master’s laser. The master acquires the slave laser and turns on its laser. The slave spacecraft acquires the master laser and phase locks with it. This process is automated and requires no ground involvement. The master spacecraft remains in contact with the ground during initial acquisition. If acquisition is unsuccessful, then the ground commands an adjustment of the fiber positioner and initiates another scan. Reacquisition does not include calibration and is autonomously performed without ground contact.

### E.7.3 Ground Station Network and Communication Parameters

The DSN 34-meter X-band antennae communicate with the master spacecraft. LISA utilizes these existing DSN capabilities at Goldstone, Madrid and Canberra (see Table E-4).

### E.7.4 Concept for Spacecraft and Science Data Destination

The LISA mission operations center is located at JPL and utilizes existing infrastructure. The mission operations center performs command sequencing, health and safety monitoring, HGA pointing, DSN scheduling, navigation and anomaly investigation.

The mission operations center sends the science data and required spacecraft data to the European and U.S. Science Data Processing Facilities (SDPF). Section D.3.4 for details of science data processing.

Figure E-5 shows the uplink-downlink data flow.

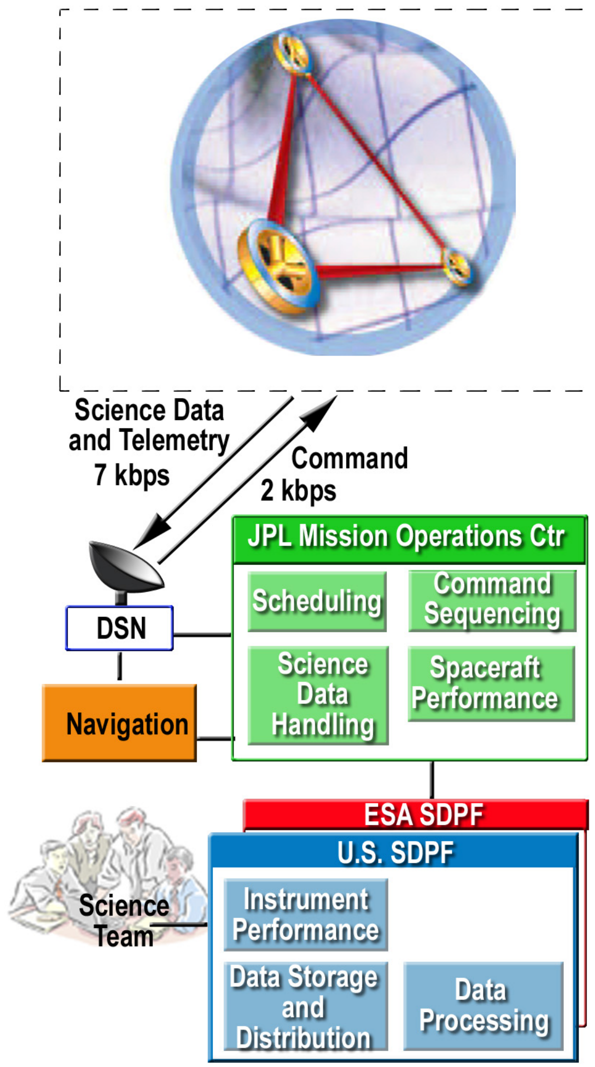



**Figure E-5: Uplink/downlink Data Flow**

### E.7.5 Uplink, Downlink Information

LISA has routine communications requirements with infrequent command uplink and a low-rate data downlink (see Table E-5). As we continue to refine our concepts for optimal sensitivity, the parameters required and the on-board processing will evolve. We can easily accommodate data volume growth by increasing the frequency of downlinks (to once per day), by increasing the on-board transmitter power, or by moving from X-band to Ka-band.

### E.7.6 Ground Software

The ground software consists of science software and JPL mission operations software adapted for LISA. The science software runs in the SDPF and performs the data analysis tasks including end-to-end error analysis and calibration. The mission operations software provides the following functions:

- Activity scheduling – maintains the timeline of activities (such as ground station contacts and HGA pointing)
- Command sequencing and uplink – integrates commands for scheduling and delivery to the spacecraft
- Health and safety monitoring of the spacecraft bus and payloads
- Spacecraft bus performance evaluation
- Navigation and tracking software
- Science data formatting – provides a file of time ordered data to the science processing functions, annotated with data quality and accounting information
- Automated telemetry analysis to detect problems and alert operations personnel
- Simulator for training and procedure checkout

### Table E-4: Communication Parameters

| Uplink Parameter | Value | Downlink Parameter | Value |
|---|---|---|---|
| Maximum S/C Distance (km) | $55 \times 10^6$ | Turnaround Ranging (Yes/No) | Yes |
| Uplink Transmitter Power (Watts) | 1000 | Required Ranging Accuracy (m) | 10 km |
| Uplink Frequency Band (GHz) | 7145-7190 | S/C Transmitting Power (Watts) | 5 |
| Uplink Transmit Antenna | 34 meter BWG or HEF | Downlink Modulation Format (Name(s)) | GMSK or T-OQPSK |
| S/C HGA Receive Gain | 24.4 | Downlink Frequency Band (GHz) | 8400-8500 |
| S/C LGA Receive Gain | 4.4 | S/C HGA Transmit Gain | 25.8 dB |
| Telecommand Data Rates (b/s) | 2000 | S/C LGA Transmit Gain | 5.8 dB |
| Telecommand Bit-Error-Rate | $10^{-6}$ | Downlink Receive Antenna | 34 m BWG or HEF |
| Frame Deletion Rate | $2 \times 10^{-3}$ | Telemetry Data Rates (b/s) | 50 bps (LGA)<br>7000 (HGA) |
| S/C Receiver Bandwidth (Hz) | 200 | Telemetry Coding (Name) | Reed Solomon & rate 1/6 convolutional |
| Telemetry Bit-Error-Rate | $10^{-6}$ | Telemetry Frame Length | 1344 bits |
| Uplink margin | >3dB | Downlink Margin (using HGA) | >3dB |
| Locations: Canberra, Goldstone, Madrid | | Downlink Margin (Using LGA) | >3dB |

### Table E-5: Uplink and Downlink Information

| Parameter | Value |
|---|---|
| Data volume (for constellation) | 672 bps science data (for constellation)<br>100 bps auxiliary instrument data (for each payload)<br>100 bps housekeeping data (for each spacecraft bus)<br>72 bps Packet overhead<br>Total (for constellation) 1344 bps<br>48 hour data volume 232 Mbits |
| On-board Storage (per spacecraft) | 256 Mbits |
| Power for communications | 5 W transmitter power, 26 bB gain |
| Downlink data rate | 7 kbps via 30 cm High Gain Antenna<br>1 bps via Low Gain Antenna |
| Data Dump Frequency | Every other day during science phase |
| Upload volume (Kbytes/day) | 1.0 |
| Uplinks per day | Every other day |
| Uplink data rate | 2000 bps |

## E.8 Approach to Ensuring Mission Success

The LISA team has already put into place a proactive risk management approach that is founded in system engineering, stems from the requirements, and is linked to the technology development effort. As shown in Table E-6, the top technical risks have been identified, assessed, and a plan for mitigation has been defined. The technology development effort is already implementing mitigation strategies for high-risk components. (See Section F for the details of the technology development effort.)

Figure E-6 shows the approach the System Engineering (SE) team uses to ensure mission success. As depicted in the figure, all aspects of the LISA mission flow from the science requirements defined in Section D. NASA and ESA use the Direct Object-Oriented Requirements System (DOORS) to capture, link and trace all requirements. Other details of the LISA requirements management process can be found in the LISA System Engineering Management Plan (SEMP) [Ref. E-1].

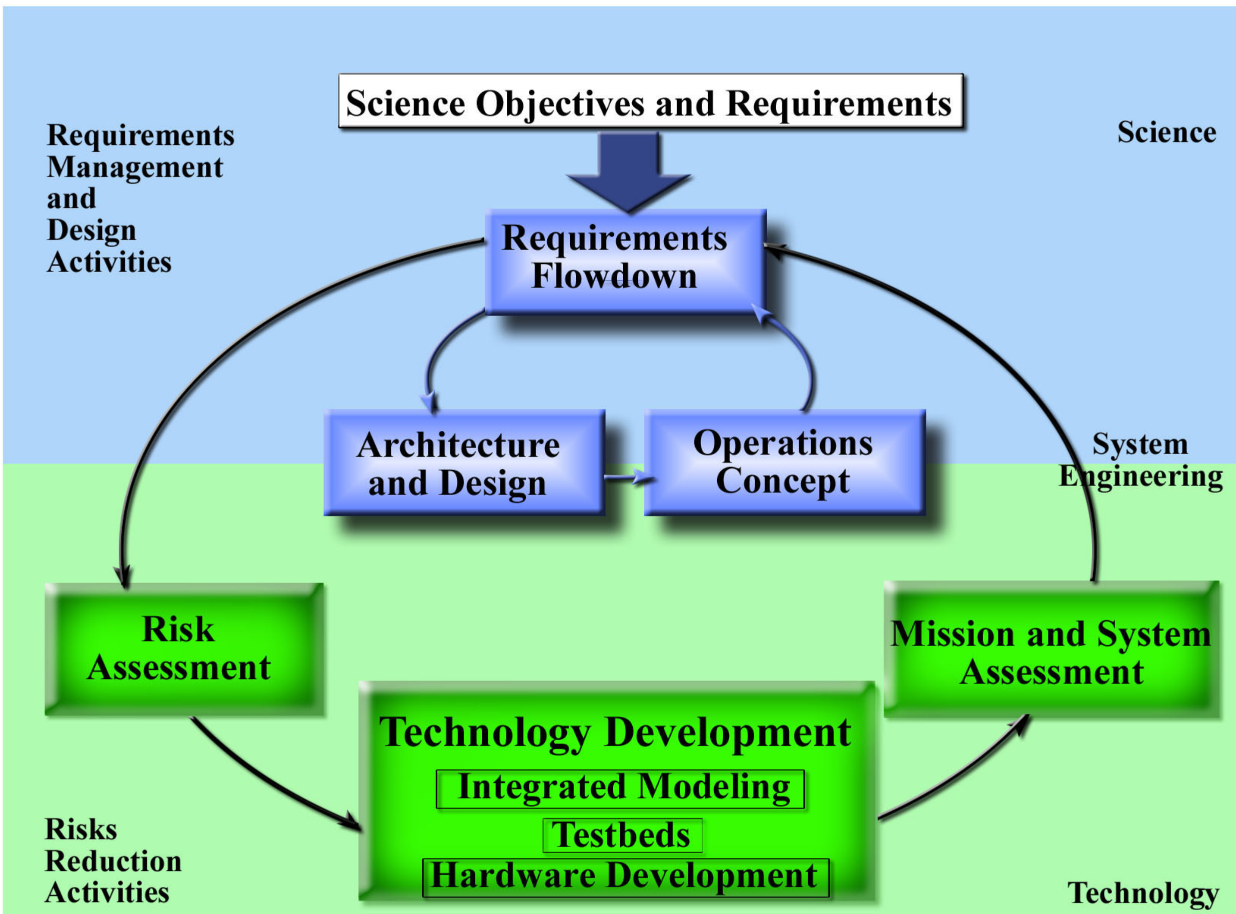



Figure E-6: The Systems Engineering Team uses a proactive approach to ensure mission success

**Table E-6: Mitigation efforts are under way for the top four LISA risks.**

| Risk Area | Impact | Mitigation | Assessment |
|---|---|---|---|
| **DRS:** The LISA GRS does not meet the requirements | Possible re-design of GRS for LISA | • Parallel GRS development with different fundamental approaches<br>• ST-7 PDR 2003, CDR 2004, SMART-2 flight demo 2006<br>• LTP PDR 2002, CDR 2003, SMART-2 flight demo 2006<br>• Extensive and targeted ground studies of suspect areas such as the study of patch fields (2002), thruster noise tests<br>• Implementation schedule allows design changes | Severe consequence, low likelihood because of mitigation |
| **IMS**: Laser does not meet the performance, lifetime, and qualification requirements | Mission degradation and performance | • Aggressive development effort involving multiple laser vendors and NASA laser qualification group<br>• Early identification of key components, and active efforts to develop and qualify needed technology.<br>• De-rating of components where applicable. Dual qualification paths on alternates where necessary.<br>• Current design incorporates redundant lasers on each optical bench.<br>• Increase the size of the telescope<br>• Shorten arm length | Moderate consequences, Low likelihood because of mitigation. |
| **DRS**: Thruster does not meet requirements | Mission degradation, performance and lifetime | • 3 parallel technology development/demonstration paths (two flight validation paths, and 1 ground test)<br>• Ground performance and lifetime test during pre-phase A<br>• Functional redundancy in design<br>• Increase gain in DRS controller | Moderate consequences, low likelihood because of mitigation |
| **System Ground Verification**: System performance on orbit does not match extrapolation of ground system test | Degradation of science | • Extensive modeling supported by test beds<br>• End-to-end system demonstration before launch<br>• On-orbit adjustment flexibility e.g. of control laws | Severe consequence, low likelihood because of mitigation |

We are currently conducting trade studies and analysis as a key aspect of risk mitigation. (The trade studies completed to date are contained in the ESA Final Technical Report (FTR) [Ref. E-2].) NASA and ESA are coordinating additional analysis and trade studies. Below is a subset of analysis and design trade studies currently under way:

- Phased array vs. gimbaled antenna
- Telescope size vs. laser power
- Redundancy vs. lifetime
- Design optimization to simplify interfaces
- Star tracker mounting
- Telescope mechanism
- Ranging and spacecraft to spacecraft communication
- Gravity balancing
- Optical bench material selection
- Strategy for integration of flight GRS
- Verification Strategy
- Location of Integration and Test

Redundancy management for LISA is handled on two levels. Traditional box-level redundancy in key components (based on a fault tree analysis performed during Phase A) is used to avoid credible single point failures in spacecraft or payload subsystems. In addition, there is the inherent redundancy in the three-arm LISA architecture. If any one arm is lost (due to failure of trans-mission or receive components on either end), the observatory still functions, albeit with slightly reduced sensitivity.

The LISA Product Assurance Program integrates proven methods and procedures at GSFC, JPL and ESA to assure the overall quality and reliability of the LISA hardware and software. During phase B, Project personnel develop a Product Assurance Plan covering all phases of system development. Drawing on established procedures in place at each project partner, the plan is tailored to unique LISA requirements.

A central feature of the LISA Product Assurance plan is the close integration of Reliability, Quality and System Safety into the overall system engineering process. This concurrent engineering approach assures overall system success through early identification of design issues. Key elements of the LISA Product Assurance effort include: reliability analyses; tracking of problems and failures and corrective actions; environmental assessment and definition of test requirements and component specifications; comprehensive effort for selection and qualification of electronic components, materials, and other system elements; contamination control planning; configuration control and hardware inspection; software mission assurance; and review of lessons-learned from previous projects.

The Integrated Systems Team (IST) conducts peer reviews throughout the project life cycle to ensure the subsystem designs meet the subsystem, system, and science requirements. Formal reviews of the requirements, architecture, design, and operations concept are conducted in accordance with NPG 7120.5.

### E.8.1 System Engineering Plan and Philosophy

The SE Office is responsible for requirements management, trade studies, and risk management. Additional information on each subject can be found in the LISA SEMP. This section focuses on the teams that support the SE Office.

The SE Office is chaired by NASA with an ESA Deputy Mission System Engineering Manager. The managers have support from the Integrated Systems Team (IST), the Systems Engineering Integration (SE&I) contractor and the Observatory Architecture Team (OAT).

Understanding the science objectives and requirements is essential to the success of the LISA mission. Therefore, in addition to the Project Scientists' role on the IST, the SE Office has established an OAT that is co-chaired by the NASA and ESA Project Scientists and is comprised of LIST members and system engineers. Its purpose is to guarantee that the emerging design fully meets the science requirements.

The IST carries out the functions of the SE Office including requirements management, verification and validation, architecture design development, operations concept development, technical resource budget tracking, risk management, interface control definition, configuration management, and technical reviews coordination (peer and project). The IST is comprised of system engineers from NASA and ESA and representatives from Science, Payload, Spacecraft, Integration and Test, and Operations.

The LISA Project Office, with support from the SE Office, selects two potential SE&I contractors in Mid-FY04, with a down-select in Mid-FY05. The SE&I deploys personnel to JPL, GSFC, and Europe to integrate LISA system engineering activities.